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Hypersonic Engine Leading Edge Experiments in a High Heat Flux, Supersonic Flow Environment

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HYPERSONIC ENGINE LEADING EDGE EXPERIMENTS IN A HIGH HEAT FLUX, SUPERSONIC FLOW ENVIRONMENT

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ABSTRACT

A major concern in advancing the state-of-the-art technologies for hypersonic vehicles is the development of an aeropropulsion system capable of withstanding the sustained high thermal loads expected during hypersonic flight. Three aerothermal load related concerns are the boundary layer transition from laminar to turbulent flow, articulating panel seals in high temperature environments, and strut (or cowl) leading edges with shock-on-shock interactions. A multi-disciplinary approach is required to address these technical concerns. A hydrogen/oxygen rocket engine heat source has been developed at the NASA Lewis Research Center as one element in a series of facilities at national laboratories designed to experimentally evaluate the heat transfer and structural response of the strut (or cowl) leading edge. A recent experimental program conducted in this facility is discussed and related to cooling technology capability. The specific objective of the experiment discussed is to evaluate the erosion and oxidation characteristics of a coating on a cowl leading edge (or strut leading edge) in a supersonic, high heat flux environment. Heat transfer analyses of a similar leading edge concept cooled with gaseous hydrogen is included to demonstrate the complexity of the problem resulting from plastic deformation of the structure. Macro-photographic data from a coated leading edge model show progressive degradation over several thermal cycles at aerothermal conditions representative of high Mach number flight.

NOMENCLATURE

c_p specific heat
 d hydraulic diameter
 h heat transfer coefficient
 k thermal conductivity
 K augmentation
 l distance from entrance
 L^* entrance effects parameter
 Nu Nusselt number, hd/k
 Pr Prandtl number, $\mu c_p/k$
 Re Reynolds number, $\rho Vd/\mu$
 T temperature
 V velocity
 θ angle
 μ viscosity
 ρ density

Subscripts

c coolant
 r wall to fluid ratio
 w wall

INTRODUCTION

With the increased interest in the United States in supersonic and hypersonic flight, a significant number of technical challenges have surfaced which are critical to the successful development of these high speed flight vehicles.

One major concern in advancing the state-of-the-art technologies for hypersonic vehicles is the development of an aeropropulsion system capable of withstanding the sustained high thermal loads expected during flight. Substantial progress has been made on computational understanding of the flow physics/chemistry and of the resulting aerothermal loads created during high speed flight. However, laminar-to-turbulent boundary layer transition, sealing high temperature articulating panels, and shock-on-shock interference heating of leading edges are especially difficult problems to solve. Consequently, there is a need for experimental facilities capable of providing a high heat flux environment for testing hypersonic vehicle component concepts and verifying (or calibrating) the analyses.

The high heat loads encountered by the leading edges of a hypersonic aircraft during flight imposes severe demands on the materials and structures which require a multi-disciplinary approach to resolve. For example, aerodynamic heating at high flight Mach numbers, including the bow-shock/shock interference heating effects on the engine cowl leading edge (Glass 1989, Holden 1988), can result in heat flux levels which exceed the capability of most conventionally cooled metallic and potential ceramic materials available for aerospace applications today. As a result, an emphasis has been placed on advancing the technology base to insure the development of unique actively cooled structures capable of withstanding these extreme environmental conditions.

An experimental research effort has been initiated at the NASA Lewis Research Center to compliment the national facilities being developed to study high heat flux in aeropropulsion systems. The focus at Lewis is to assess the capability of actively cooled structures to tolerate the high heating rates typical of hypersonic flight (Gladden and Melis 1988, 1993, Gladden et al. 1990, Melis et al. 1989). The "Hot Gas Facility", a hydrogen/oxygen rocket engine with a 0.9 cm square cross section combustion chamber, provides hot combustion gases up to 3050 K and 4100 kpa to the test articles. A convergent-divergent (C-D) nozzle can be attached to the combustion chamber to achieve supersonic flow of up to Mach 2.5. Test specimens can be mounted in the combustion chamber itself or down stream of the C-D nozzle exhaust to obtain either subsonic or supersonic flow conditions. Heat flux levels up to 450 w/cm² on the side wall and up to 5,000 w/cm² at the stagnation point can be obtained in a supersonic flow field.

An oxidation/erosion experiment on protective coatings is presented herein to demonstrate the facility capabilities. The objective of the experiment is to investigate the survivability of a coating on a simulated leading edge configuration in a high velocity, hot gas environment. Specifically, two areas are addressed: 1) to determine if

high velocity hot gas impingement on leading edges will cause erosion of the heat exchanger face sheet protective coating in a reducing (fuel rich) environment, and 2) to assess the effect of pre-oxidation of the coating on its (rate of) erosion in an oxidizing (oxygen rich) flow. This experiment was guided by both thermal and structural analyses.

Results from this experiment in a supersonic flow environment and some representative analyses are presented. These data will guide the design and development of actively cooled structures exposed to high heat flux in aeropropulsion systems.

BACKGROUND

Vehicle flight at hypersonic speeds in the atmosphere at high dynamic pressure presents significant heat transfer challenges throughout the propulsion system and on the airframe because of the high aeroheating loads. Both the magnitude and the uncertainty of these heat loads is of concern. For instance, if there is an early boundary layer transition on the inlet forebody, then an active cooling scheme may be required. However, if boundary layer transition occurs far downstream, then a passive cooling may be acceptable. In addition, ram/scram type engines have articulating panels to accommodate a wide range of flight Mach numbers. These high temperature moving panels must effectively seal the engine flow path and prevent exposure of the support structure to the high temperature gases. Heat flux levels on the order of 2,000 w/cm² in the inlet duct and combustion chamber make active cooling of the panels a necessity and active cooling of the seals a distinct possibility.

Much higher heat fluxes can occur in high speed flight when there is a shock-on-shock interaction at a cowl or strut leading edge. As the vehicle accelerates through the atmosphere the airframe shock sweeps across the engine inlet creating an interaction with the bow shock on the engine leading edge component (fig. 1). The physical phenomena of this interaction are reasonably well understood and can be divided into six classes of interaction which were identified by Edney (1968) (fig. 2). The central portion of figure 2 show a representative cowl leading edge with a characteristic bow shock (BS). Also shown is an oblique impinging shock (IS). The type of interaction is dependent on the circumferential location of the interaction. Sketches of the six types are shown around the leading edge. The most severe heating occurs with the type IV interference heating phenomena. Holden et al. (1988) and Glass et al. (1989) have measured stagnation point heat transfer augmentation factors of more than 20:1 for type IV interference heating phenomena. Local heat flux levels in the stagnation region of the engine cowl and

struts range up to $10,000 \text{ w/cm}^2$ for some typical flight conditions without shock interference heating. With shock interference heating, local stagnation heat flux can be amplified to values of $50,000$ to $60,000 \text{ w/cm}^2$. This high heat flux occurs within a narrow band which sweeps across the structure as the vehicle accelerates. A typical schlieren photograph of a type IV interference pattern is shown in figure 3 and clearly depicts the supersonic jet that impinges on the leading edge. Navier-Stokes real gas analysis has been able to capture most of the flow physics and chemistry of this phenomena. However, the challenge is finding methods to reduce the heat load and/or finding cooling schemes and materials that can tolerate this environment.

HOT GAS FACILITY and OPERATING CHARACTERISTICS

An experimental research facility was developed at the NASA Lewis Research Center to compliment the national facilities being developed to study high heat flux in aeropropulsion systems. The focus at Lewis is to assess the capability of actively cooled structures to tolerate the high heating rates typical of hypersonic flight.

General description

The Hot Gas Facility can provide Reynolds number, Prandtl number, enthalpy, and heat fluxes similar to those experienced during hypersonic flight without shock interference heating. When operated in the oxygen rich mode, the atmospheric partial pressure of oxygen can also be simulated. Hydrogen/oxygen combustion gases ranging in temperature from about 1300 K to about 3050 K at combustion chamber pressures up to 4100 kpa are achievable. The products of combustion are water vapor and either hydrogen or oxygen depending on whether the test is run fuel rich ($\text{O/F} < 8$) or oxygen rich ($\text{O/F} > 8$). The Prandtl number for these mixtures is in the range of 0.6 to 0.8 which is comparable to air. In addition, the ratio of specific heats is in the range of 1.2 to 1.5 which is also comparable to that of air. Reynolds (Re/l) number similarities can be maintained up to $1,000,000$ per meter.

The operating characteristics of the facility are shown in figure 4 and related to the flight characteristics of a high speed flight vehicle. These characteristics are plotted as chamber pressure versus fuel-air mixture ratio which are the controllable parameters in the facility. Both the erosion and the oxidation envelopes for the facility are shown in dark shading. The X-30 flight conditions are shown in the cross-hatched shading. The partial pressure of Oxygen, the mass velocity and heat flux are the significant parameters that determine the boundaries both the flight vehicle and the test facility. As noted in the figure, these parameters are coincident for the oxidation tests and the X-30 flight.

However, only the mass velocity conditions are matched between the erosion tests and the X-30 flight while the heat flux values are similar ($3,000$ vs $4,000 \text{ w/cm}^2$) but not matched. (Even though not shown, points of constant mass velocity are nearly horizontal in the lower portion of the figure. Constant heat flux values form a bucket shaped curve facing up with a minimum near an O/F of 8 and constant oxygen partial pressure points are nearly vertical.)

The rocket engine combustion chamber, mounted horizontally, has a square cross section of 0.9 cm . The test stand and the exhaust scrubber tank inlet pipe are shown during firing in figure 5. The rocket engine is located to the right and is firing from right to left in the figure.

Gaseous hydrogen coolant is used for these test articles in the facility. A liquid nitrogen heat exchanger is also available to chill the coolant to a temperature of 90 - 100 K . Hydrogen coolant can be supplied to a test specimen at a maximum flow rate of 0.068 kg/sec and a maximum pressure up to $10,000 \text{ kpa}$.

In general, the facility is operated in short bursts of about five or six seconds. However, tests as long as 30 seconds have been conducted with no damage to the facility hardware. A convergent/divergent (C-D) nozzle with a 1.85 area ratio can be added to provide a supersonic Mach number of about 2.0 . With a nozzle extension attached, a Mach number of about 2.6 can be attained.

Data acquisition capabilities

TRADAR 2.5, a high speed, analog-to-digital recording system is used to record up to 100 data channels at 100 samples per second per channel of time dependent data. Data from pressure, temperature, flow, and strain sensors can be recorded for subsequent computer processing. A 32 -channel recording oscillograph is used to provide immediate data feedback on critical data channels in any given test. Twenty parameters can be recorded in this fashion. There are 48 data channels for each of the type K, type E, and type R thermocouples. Sixty four channels of signal conditioners as well as 12 frequency-to-voltage converters are available for use with pressure transducers. For photographic documentation, the facility has high-speed motion picture cameras which operate at up to 400 frames per second as well as video and 35 mm remotely operated cameras.

Leading edge configuration

The initial leading edge coating erosion/oxidation specimen is shown in figure 6 mounted on the exhaust jet center line of the rocket engine. The view is looking upstream at the exhaust nozzle. This specimen has an external shape representative of a cowl leading edge, however, the internal coolant passage is a simple drilled

hole. Phase 2 test specimens had a sophisticated convection cooled channel flow design. For the experiments discussed, the C-D nozzle is installed without the nozzle extension to yield a Mach 2.0 condition at the test specimen leading edge. This is an under expanded flow condition. However, it provides similar wall shear conditions as that expected in actual flight.

Metallurgical tests are performed periodically between the thermal cycling experiments to track the behavior of the coating. Two specimens are tested in a reducing (fuel rich) environment and two specimens are tested in an oxidizing (oxygen rich) environment. Each of these specimens are subjected to thermal cycling with a goal of about 150 seconds total exposure.

The heating rates on a typical leading edge model are determined from a heat transfer calorimeter specimen fabricated from titanium aluminide. The calibration specimen is subjected to hot gas flow for about two seconds at an O/F ratio ranging from 1.17 to 1.7. Data from these tests are reduced by solving the transient wall temperature as a response of a semi-infinite body to a step change in boundary conditions. The stagnation heat flux levels determined by this method are consistent with that predicted by correlations in the literature for cylinders in cross flow.

NUMERICAL PROCEDURES

Thermal and structural analyses are performed on components tested in the hot gas facility to predict temperature and stress profiles on the test piece. PATRAN, a finite element pre and post processor is used to create the analysis models. PATRAN translates the models as input for the thermal and structural analysis codes. A finite difference technique using the analysis program SINDA is used to predict the thermal response of a component as it undergoes testing in the Hot Gas Facility. The temperatures from SINDA are then mapped onto the model as thermal loading for structural analysis using MARC, a finite element program.

Shown in figure 7 is a typical example of an actively cooled cowl leading edge (Gladden 1990). The coolant follows a rectangular channel, as shown by the arrows, around the leading edge to remove the heat. This particular model was used to evaluate the shock impingement phenomena, hence, there is a higher density of elements in the region where a type IV impingement occurs in order to effectively model the high thermal and stress gradients. Initial conditions and constraints required for the analyses are determined as follows.

Aerothermodynamic Heat Flux

This configuration is assumed to be operating the flow

field and subjected to the external heat flux environment described in the background section. The heat flux imposed includes a simulated shock-shock interference heating effect described by Glass et al. (1989) and Holden et al. (1988). This heat load is very intense (approx. 50,000 w/cm²) but over a narrow width of about 1/20th of the leading edge diameter.

Coolant Correlations

The emphasis of the analysis is to demonstrate the capability of actively-cooled structures to survive in a hostile environment with an extremely high, localized heat flux. A convection cooling scheme enhanced by curvature (Hendricks and Simon, 1963) is considered for this analysis.

$$Nu = .025 K(\theta) Re^3 Pr^4 T_r^* L^* \quad (1)$$

where $K(\theta)$ is an augmentation factor which is a function of the channel curvature.

$$T_r = T_w/T_c$$

$$L^* = (1.0 + 0.3(1/d)^{.7})$$

$$\text{and } n = -0.55$$

The correlation is modified by the coolant-to-wall temperature ratio, T_r^* , (Hendricks et al. 1979) to account for large temperature differences that exist between the fluid and the surface of the structure.

RESULTS AND DISCUSSION

Convection-Cooled Model

The results of the analyses are presented for a convection-cooled configuration and consists of a characteristic temperature distribution and the resulting equivalent plastic strain due to thermal gradients. Figures 8 and 9 depict typical results from the analyses. For the case shown, the material selected was tungsten. It can be seen that wall temperatures at the shock-on-shock interaction location are near 1800 °K which is within typical operating limits for tungsten. Plastic deformation occurs at the shock interaction region and shows the maximum equivalent plastic strain to be around 0.7%.

Coating erosion/oxidation experiments

All four test articles were tested beyond the original time/cycle goal at erosive and/or oxidizing conditions representative of high speed flight. The oxidation/erosion coating performed well during the erosion tests. No coating cracking or spalling was detected during the thermal shock

and thermal cycle tests.

During the oxidation tests, coating degradation was observed to take place with increasing thermal cycles. Figure 10 depicts this progressive process with photos taken after each series of tests run in the facility. There is some discoloration of the coating after the first cycle. The discoloration/oxidation of the coating grew more pronounced after each series of thermal cycles.

Data taken from these tests, in conjunction with data from other national laboratories, are being compiled for use in design decisions regarding coating development and application on high speed flight hardware.

CONCLUDING REMARKS

A high heat flux experiment related to a high speed flight environment is discussed. The four leading edge coating erosion/oxidation test articles were tested beyond the original time/cycle goal at conditions representative of high speed flight. The coating performed well during the erosion tests. No coating cracking or spalling was detected during the thermal shock and thermal cycle tests. During the oxidation tests, the coating degraded progressively throughout the test series.

Detailed thermal/structural analysis was performed on an actively-cooled concept similar to that which might be applied as leading edge configuration in a hypersonic flow field shock-on-shock interference heating. This concept relied on enhanced internal cooling using hydrogen as a coolant and highly conductive materials to spread the incident heat flux over a large area. There is, however, a significant level of uncertainty in the internal film coefficients for the Reynolds numbers considered for this application. Further study in this area, including three-dimensional effects, is desirable.

The combination of high internal film coefficients, which may be attained with enhanced convection cooling, and the high thermal conductivity of a copper alloy keeps the maximum wall temperature within acceptable limits. However, the large thermal gradients result in a noticeable amount of equivalent plastic strain (and deformation) in the leading edge region. With this in mind, it is necessary to further investigate the behavior of these structures by incorporating mechanical, cyclic, and time dependent effects. With plastic strain occurring at high temperature, a number of other phenomena can take place such as creep ratcheting and/or creep buckling where small amounts of plastic strain can accumulate over a period of time or number of cycles and could ultimately result in failure of the structure. These mechanisms, as well as life prediction,

need to be considered in more sophisticated analyses. The surface protective coating must be able to accommodate these deformations for the required number of operating cycles.

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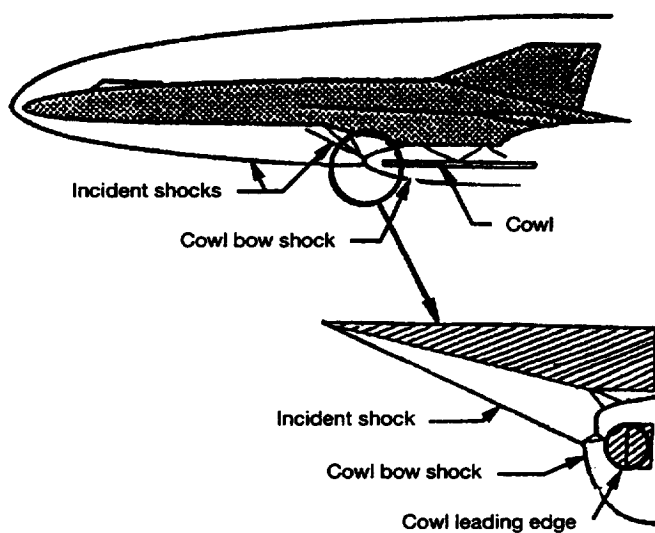


Figure 1.—Shock-on-shock interference heating of leading edge.

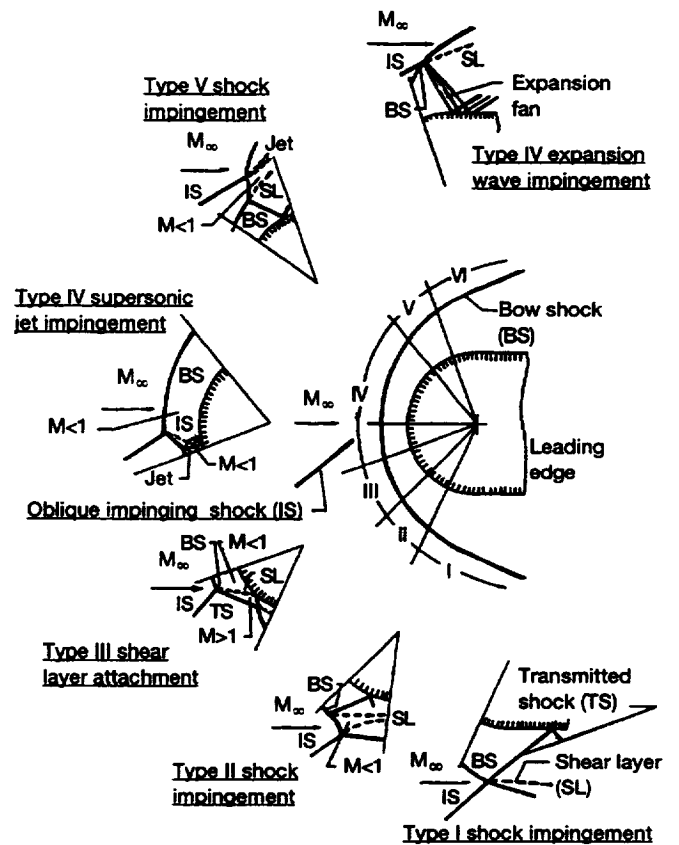


Figure 2.—Six types of shock interference patterns and their location.

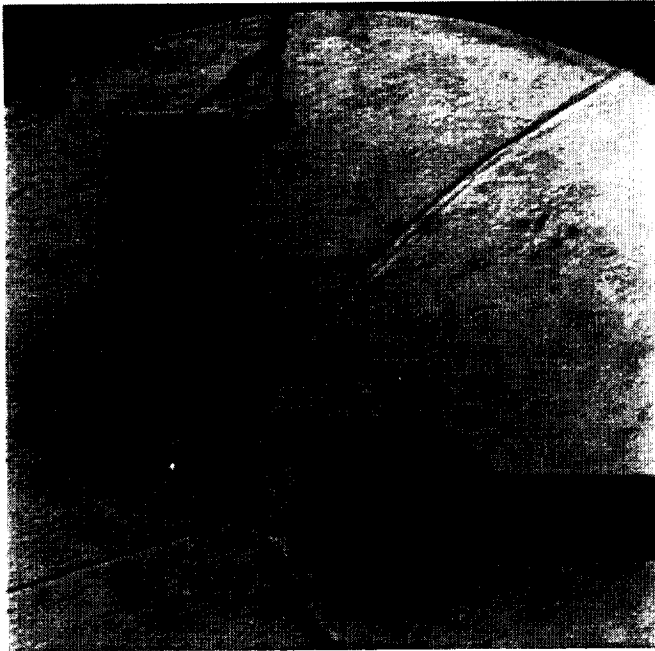


Figure 3.—Schlieren photograph of type IV shock interference pattern.

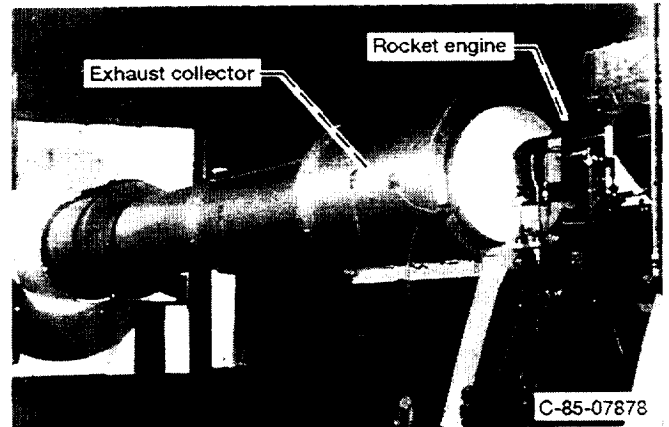


Figure 5.—Hot gas facility during firing of the rocket engine.

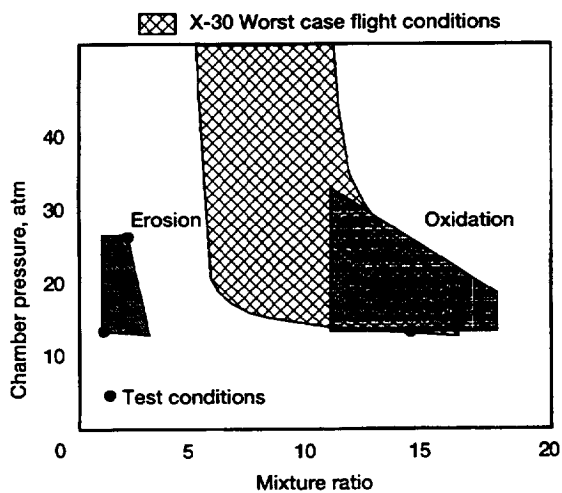


Figure 4.—Comparison of high speed flight environment and the hot gas facility test conditions.

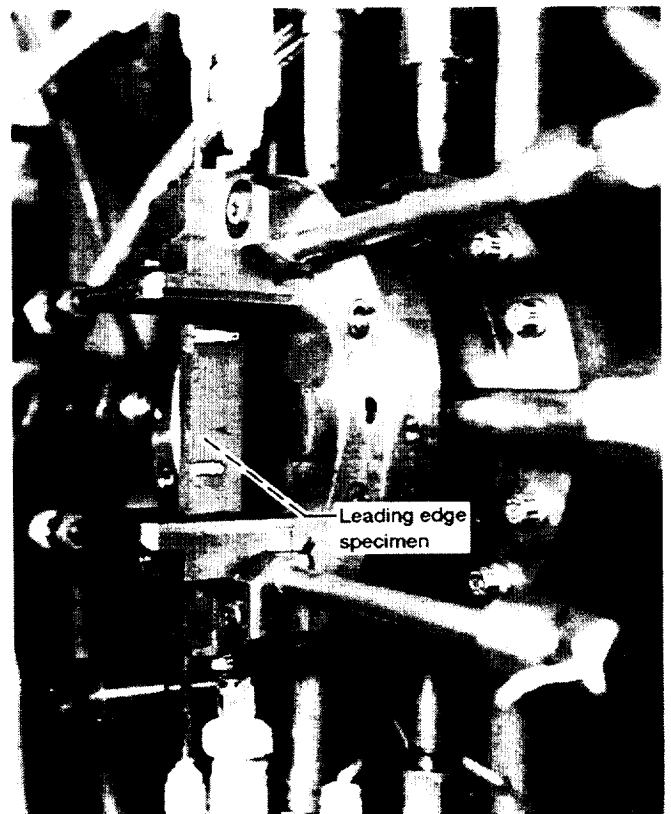


Figure 6.—Coated leading edge specimen mounted at C-D nozzle exhaust plane.

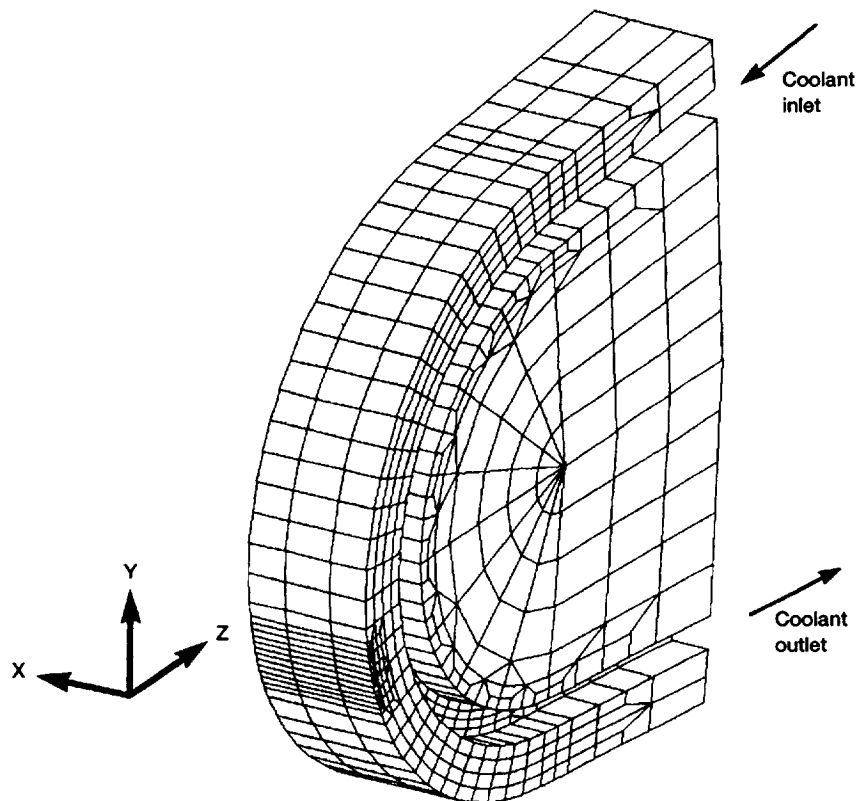


Figure 7.—Finite element model of a convection-cooled leading edge.

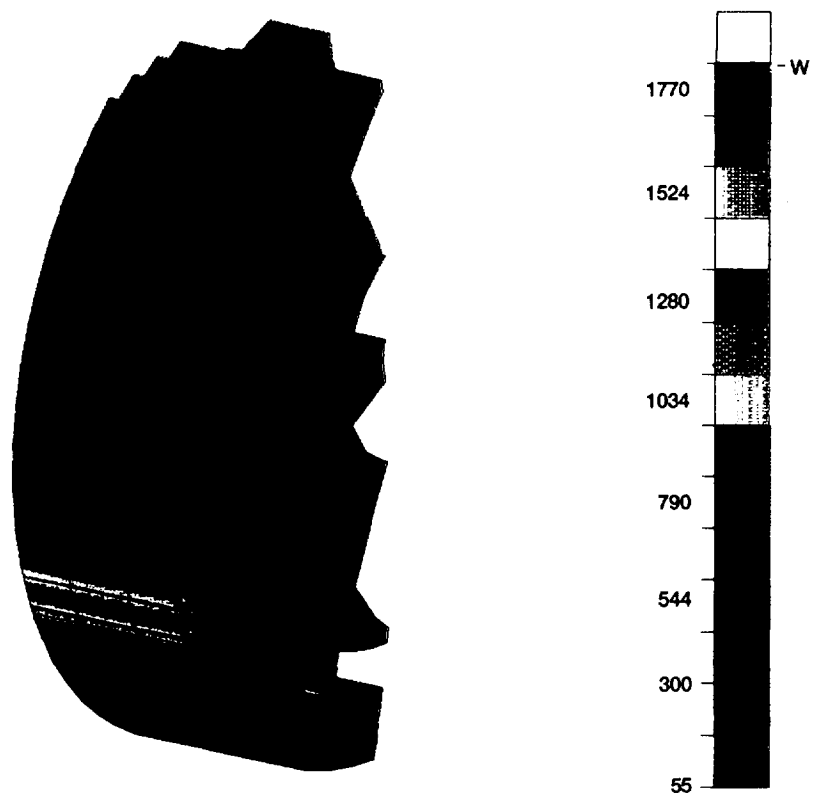


Figure 8.—Temperature distribution °K with heat flux spike.

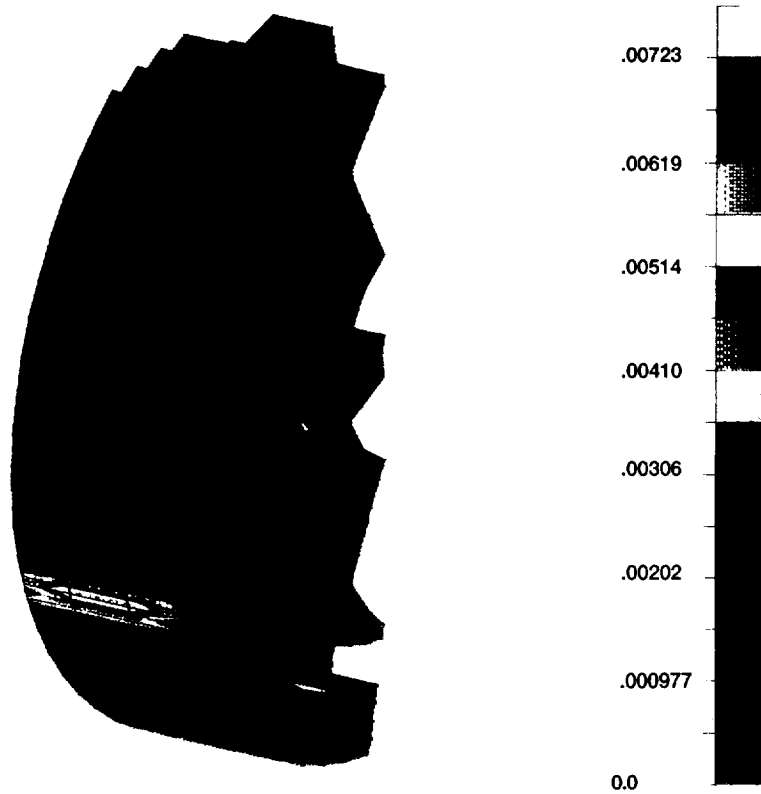


Figure 9.—Equivalent plastic strain distribution.

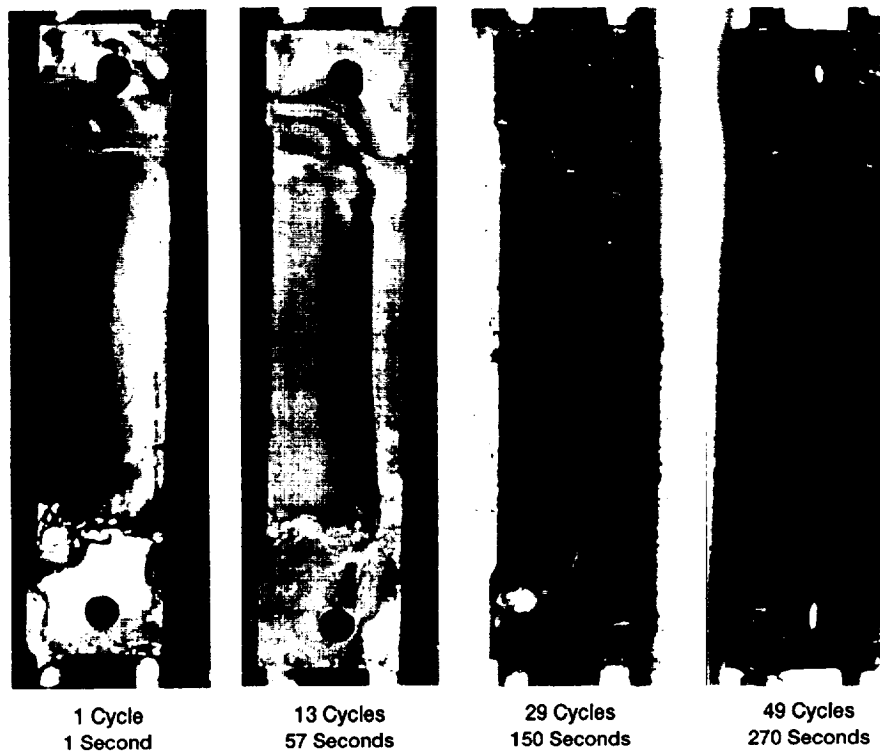


Figure 10.—Macro photographs taken during oxygen rich tests shows coating degradation.

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